4. LAUNCH AND ORBITAL OPERATIONS [1]

4.1 PHASES OF LAUNCH THROUGH ORBITAL OPERATION

Launch and orbital operations can be divided into two or three phases:
(1) the initial launch and boost phase which terminates when the vehicle obtains the velocity and altitude necessary to achieve Earth orbit;
(2) the orbital transfer phase, during which properly timed firings of rocket motors move the satellite into the desired final orbit; and
(3) depending upon the mission, return from orbit. Re-entry is further discussed in Vol. 2, Ch. 7.

4.1.1 Launch Phase

The prime objective during the launch phase is for the boost vehicle to overcome Earth's gravitational pull, rise through the atmosphere and overcome frictional heating. It must provide a satellite with an initial vertical and final orbital velocity (almost parallel to the surface of the Earth) using sustainer and upper rocket stages which will keep it in orbit. Depending on the latitude of the launch point, the desired orbital inclination and altitude, the initial orbit may not be the final orbit for the satellite. To change inclination the boost and higher stages of the ELV must rotate the attitude of the vehicle, so that it will be moving in the proper direction, and then pitch over to the orbital plane gradually as it gains velocity and altitude.

The gradual programmed pitchover (called a gravity turn) is carefully designed so that the angle of attack (the angle between the axis of the vehicle and the vector of the aerodynamic forces) is kept as close to zero as possible. The gravity turn is preceded by a small pitchover maneuver called the "kick angle." If this is not accomplished, the aerodynamic loads on the vehicle will build up and overcome the guidance and control system, thereby producing a deviation from the planned flight path. If the angle of attack becomes too large, the airloads may over-stress the vehicle and cause its structural failure. The aerodynamic force effects are proportional to one half of the product of the local atmospheric density and the vehicle velocity squared (called dynamic pressure or "q"). In some vehicles, failure can begin at less than 10 degrees angle of attack during

[1] The information in this chapter was developed using the references listed at the end of the chapter. This material is intended for readers with little or no background in either orbital mechanics or rocketry. Others can proceed directly to Chapter 5.

the "high q" portion of flight. A typical trajectory profile is shown in Figure 4-1.

When the vehicle reaches a very high altitude, the atmospheric density becomes so low that the dynamic pressure is essentially zero regardless of the velocity. After this, the zero angle of attack is no longer required and different pitch attitudes and pitchover rates can be used.

Control of all launch vehicles is maintained by gimballing (tilting) the engine nozzles or some equivalent way for changing the direction of the engine thrust. Launch vehicles must be controlled continuously because they are, without exception, aerodynamically unstable, i.e., a slight increase in angle of attack will cause the aerodynamic forces to attempt to increase the angle of attack even further. Severe wind shears during the early post-launch period of flight create difficulty for most vehicles, as the guidance and control systems must act to minimize the pitching or yawing due to abrupt angle-of-attack changes which they create.

Most launch vehicles contain several stages. Thrust is initially provided by the lowest (and largest) boost stage. When the fuel for this first stage is consumed, the spent fuel casing is jettisoned to Earth, the remainder of the vehicle separates from it and the next stage is fired to continue the flight.

Part of the preparation for any mission is the planning for the impact location of the spent stages (and other jettisoned equipment) in order to minimize the risk to people and property on the ground (see Ch. 2).

Most of the current launch vehicles use solid rockets fastened to a central core vehicle which is usually a liquid propellant stage. These "strap-on" solid rocket motors (SRM's) augment the first stage thrust and are jettisoned when their propellant is consumed.

4.1.2 Orbital Insertion and Orbital Operations

It is not possible to describe the myriad of possible orbital parameters which may be desired or designed for different mission objectives. This discussion will only briefly cover the very simplest example. Consider the sequence of events illustrated in Figure 4-2. In the first illustration (a), a satellite (with a booster stage) is placed in a low "parking" orbit around the Earth. The rockets are fired in orbit and then shut off. The result of this orbital correction firing is the creation of a new elliptical "transfer" orbit which has an apogee (greatest distance from the Earth) which is at a higher altitude above the Earth than the original orbit (Figure 4-2(b)). If the satellite has no further propulsion, it will continue to follow this

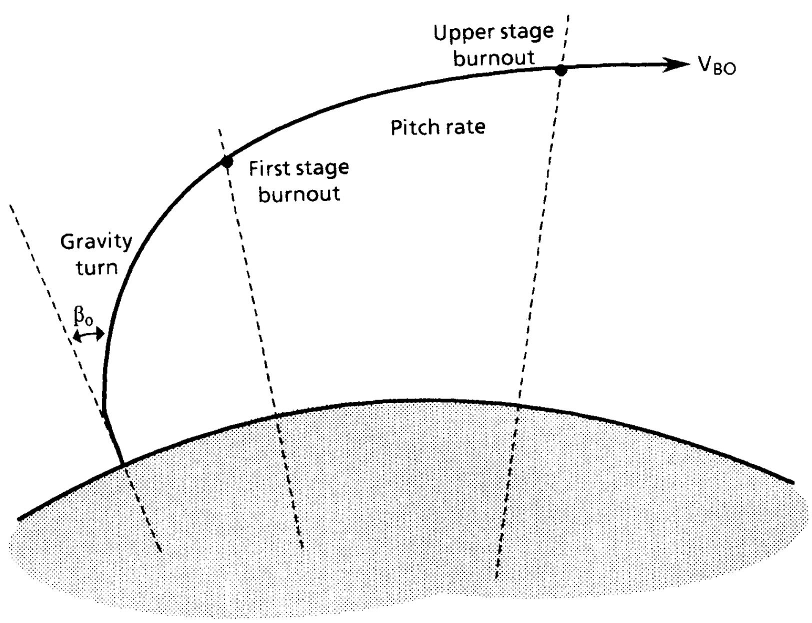


FIGURE 4-1. TYPICAL TRAJECTORY PROFILE

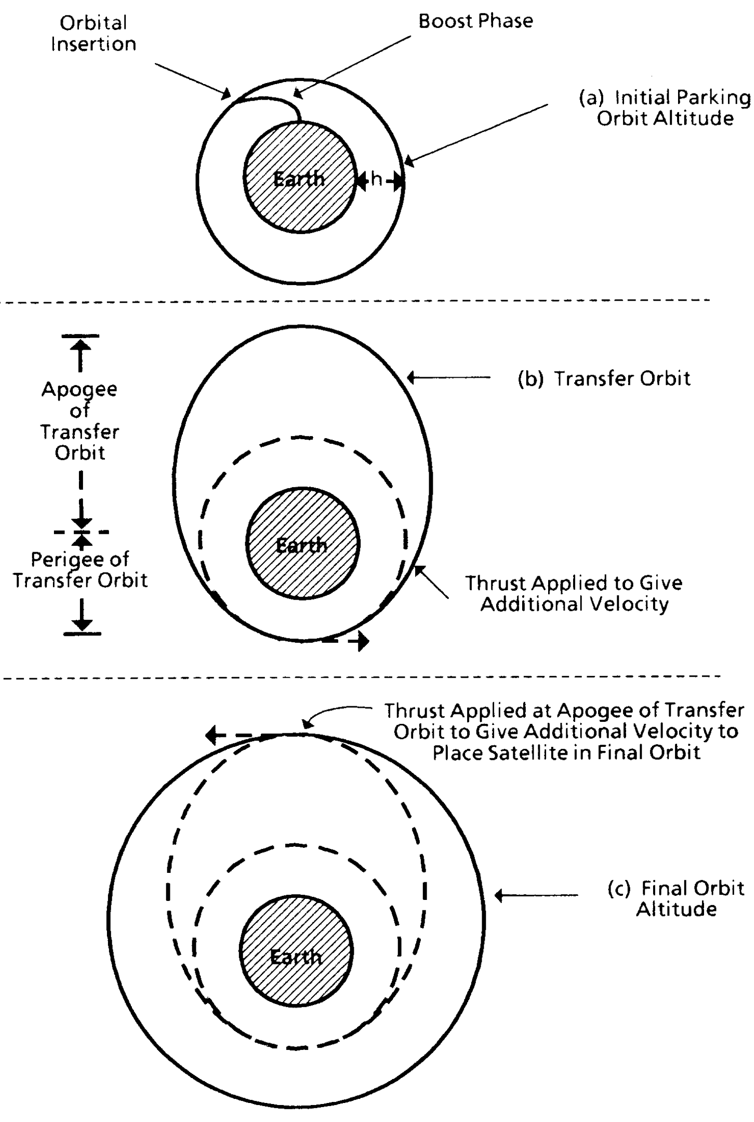


FIGURE 4-2. ORBITAL TRANSFER

elliptical orbit indefinitely, passing (ideally) through its initial perigee point once very revolution. If the objective is to reach a higher circular orbit, the built-in rockets (apogee kick motors, AKM) can be fired again (for a specified period of time) when the satellite reaches the apogee of the elliptical orbit, and the new orbit will be as shown in Figure 4-2(c).

4.1.3 Orbital Decay and Re-entry

Once out of the densest portion of the atmosphere, the ELV and its payload (satellite) has only very small drag forces acting upon it to reduce the satellite velocity. Consequently, the satellite will continue to orbit until reverse thrust (retro-propulsion) is applied for a planned re-entry or decay forces eventually cause an uncontrolled re-entry. Controlled descent from an orbit reverses the firing sequence for orbit transfer. Rocket engines fire for a determined interval and angle and the vehicle/satellite now follows an elliptical orbit with apogee at the original orbital altitude and perigee at an altitude much closer to the Earth. If the perigee is within denser portions of the atmosphere, the vehicle/satellite will start to slow down gradually because of aerodynamic drag and descend to the Earth sooner due to orbital decay (see Vol. 2, Ch. 7). Aerodynamic heating is intense because of the very high vehicle velocity as it is coming out of orbit and the slow initial braking during re-entry. Objects not designed to withstand this heat by protection from a heat and ablation shield generally break up and, often, vaporize altogether. Re-entry vehicles (RV's) similar to those provided for ICBM's have been proposed for recoverable payloads.

Satellites which are placed in very low Earth orbit may not need any propulsion to return from orbit. Even at an altitude of 200 miles, the very low density of air molecules still applies a small, but continuous drag force. These satellites will very slowly lose both velocity and orbital altitude and the decay will gradually increase until the object is traveling slow enough to re-enter the Earth's atmosphere. This unplanned re-entry is discussed further in Ch. 7. Figure 4-3 shows approximate orbital lifetimes for satellites in circular orbits. Orbital lifetime is a direct function of the mass to drag ratio of the satellite. This ratio is represented by the ballistic coefficient β which is equal to W/C_DA; where W is the weight, C_D is the drag coefficient of the body, and A is the cross-section area. The shaded area in the figure shows the range of lifetime in orbit for objects whose ballistic coefficients range from 10 to 300 lb/ft^2. The larger values of ballistic coefficient correspond to the longer lifetimes in the shaded region shown in Figure 4-3.

If rocket engines are used to de-orbit, as proposed for recoverable payloads that use re-entry vehicles (RV's), the

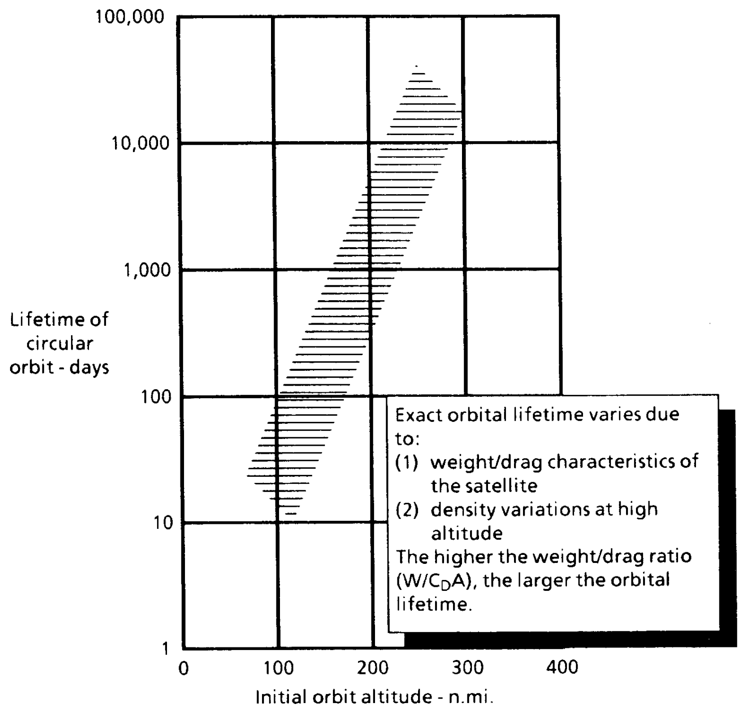


FIGURE 4-3. APPROXIMATE LIFETIMES FOR SATELITES IN CIRCULAR ORBITS

potential hazard from the re-entering spacecraft is controllable. However, the hazard from a decayed satellite re-entry is uncontrolled and usually cannot be predicted with any accuracy (see Ch. 7).

4.2 BASIC ORBITAL CHARACTERISTICS

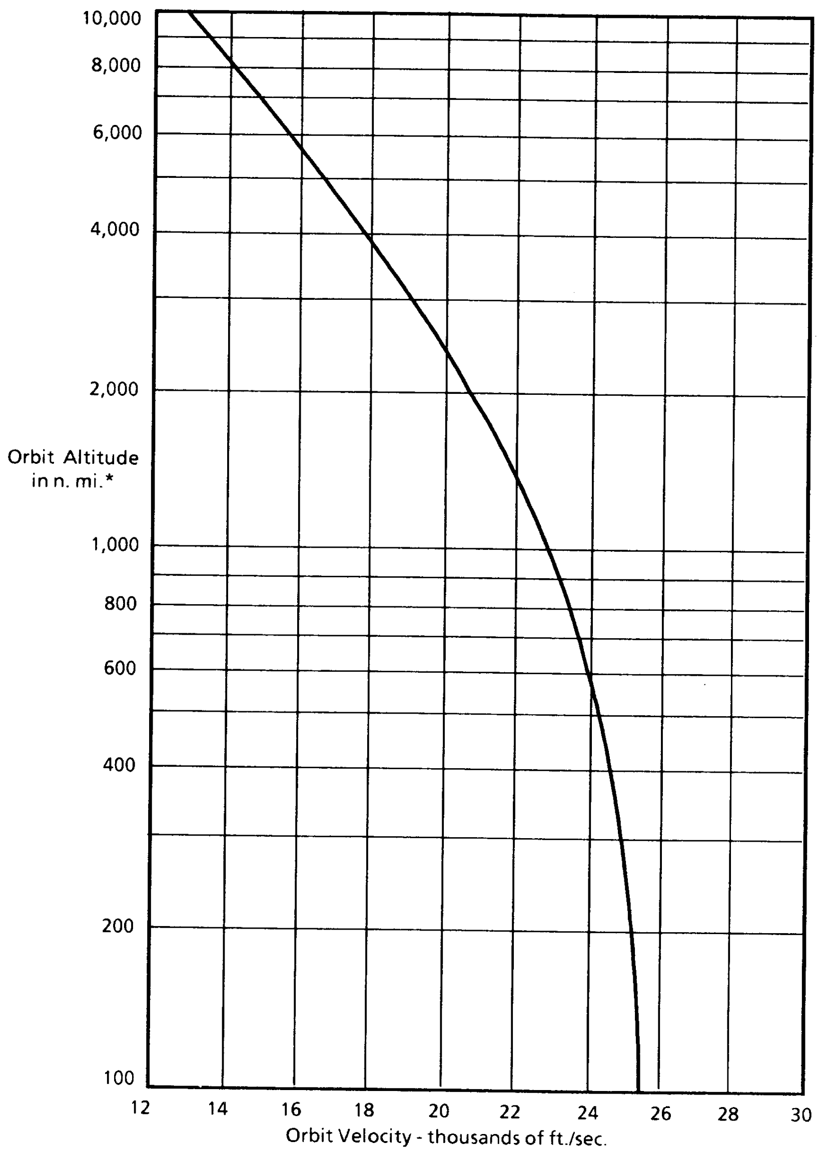
A satellite stays in orbit because the centrifugal (outward) force equals the Earth's gravitational pull (inward). The centrifugal force is proportional to V^2/R, where R is the distance from the center of the Earth to the satellite and V is the component of satellite velocity which is perpendicular to the radius R. The gravitational pull decreases with distance and is proportional to $1/R^2$. For low Earth orbits (LEO), the gravitational pull is stronger and, consequently, satellites must have a higher velocity to compensate and, thus, circumnavigate the globe much more rapidly. Figure 4-4 shows the relationship between orbital velocity and altitude above the surface of the Earth for circular orbits. Figure 4-5 gives the period (the time required to complete one circular orbit) as a function of altitude above the surface of the Earth.

Not all orbits are circular; many are elliptical and are employed in orbital transfer and other mission applications. The perigee of an elliptical orbit is the minimum altitude of the orbit; the apogee is the maximum altitude (see Figure 4-6). The eccentricity is a measure of the ellipticity of the orbit. The formula for eccentricity is:

$$\mathbf{e = r_a - r_p \div r_a + r_p} \qquad (4\text{-}1)$$

where r_a is the distance from the center of the Earth to the apogee altitude and r_p is the distance from the center of the Earth to the perigee altitude. The apogee and perigee altitudes for a circular orbit are equal, hence a circular orbit has zero eccentricity. Elliptical orbits having the same perigee altitude as a circular orbit always have a longer period, with the period increasing with the eccentricity.

The free flight path of a suborbital rocket or an expendable launch vehicle (ELV) is also elliptical. These vehicles, after completion of powered flight, follow a ballistic trajectory with an apogee above the surface of the Earth and a perigee which is below the surface of the Earth (see Figure 4-6).



* A **nautical mile** (6076.166 feet) is a measure commonly used in orbital mechanics. There are 60 nautical miles to a degree of latitude. In contrast, a statute mile has 5280 feet or 1760 yards

FIGURE 4-4. SATELLITE VELOCITIES IN CIRCULAR ORBITS

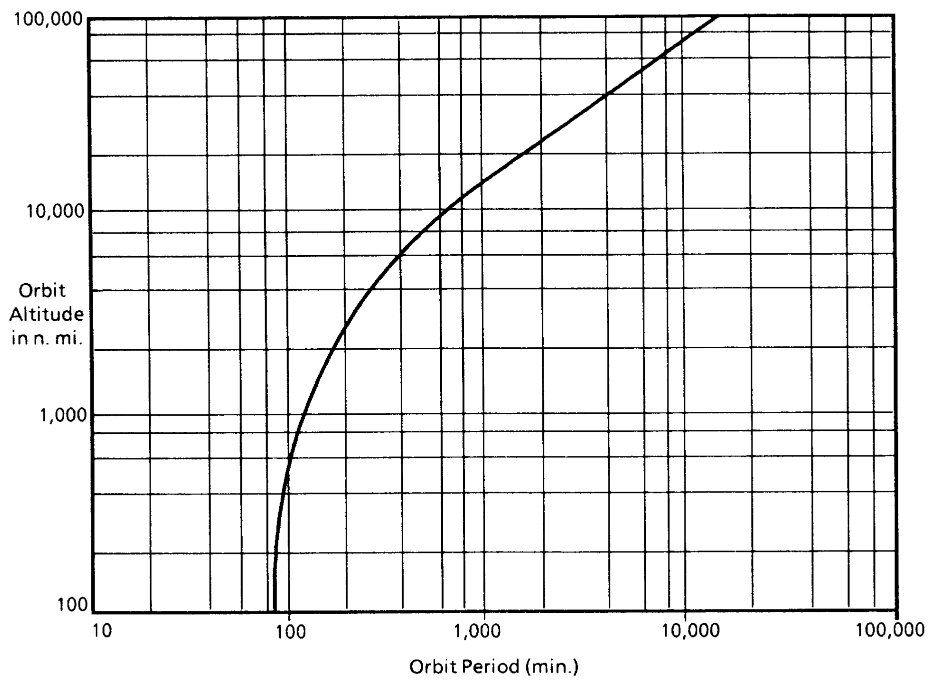


FIGURE 4-5. PERIODS FOR CIRCULAR ORBITS

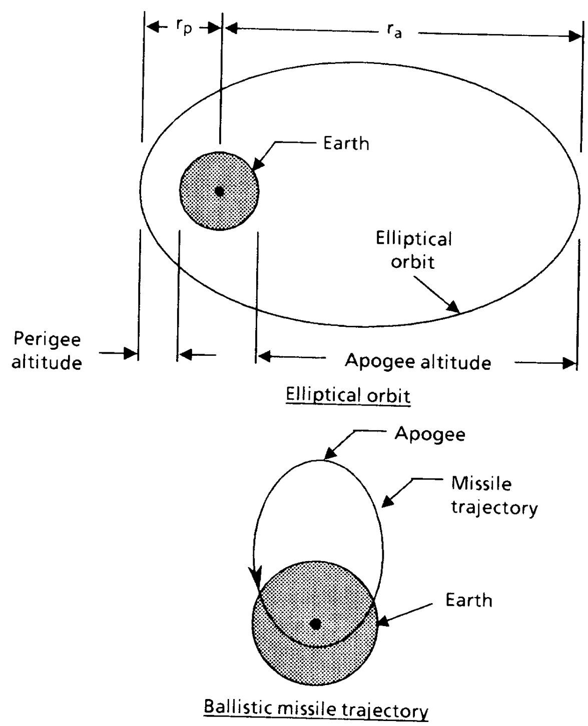


FIGURE 4-6. GEOMETRY OF SATELLITE ORBITS AND ELV TRAJECTORIES

The concepts of energy and angular momentum are essential in understanding orbital mechanics. The total mechanical energy has two components, kinetic energy (K.E.) and potential energy (P.E.). As long as no additional force is being applied to the satellite (e.g., aerodynamic or rocket thrust), the total energy of the satellite remains constant, i.e.,

Total mechanical energy = K.E. + P.E. = constant (4-2)

The kinetic energy is proportional to the square of the velocity of the satellite. Potential energy results from the combination of gravitational attraction and distance to the gravitational source. The total energy per unit mass, E, will remain constant throughout the orbit (circular or elliptical) unless a force impulse, such as rocket thrust or drag, is applied to the satellite. Thrust in the direction of the velocity vector will increase the energy and thrust or drag in the direction opposite to the velocity vector will decrease the energy.

Hence, an orbiting satellite has both Kinetic Energy: $KE = mv^2/2$ and Potential (Gravitational) Energy: GmM/r at its orbit altitude ($r = R+h$; where R is the Earth's radius, h is altitude above the Earth and M is the Earth's mass). The constant $\mu = GM$ in $(ft/sec)^3$ or $(m/sec)^3$ is the constant product of the Universal Gravitational constant and Earth's mass.

This simplifies the total energy per unit mass for an orbiting satellite to a specific mechanical energy:

$$E_s = E \div M = (KE + PE) \div m = (v^2 \div 2) - (\mu \div r) = \text{const.} \quad (4\text{-}3)$$

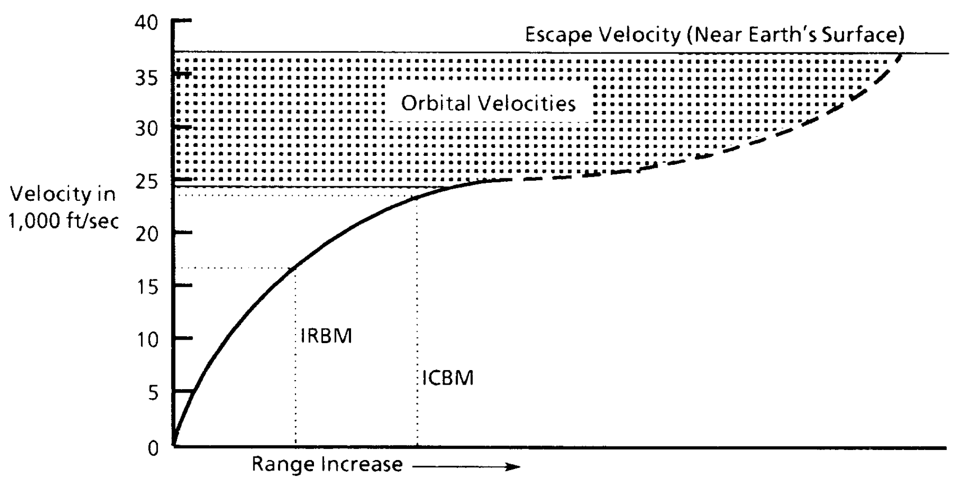
If $E_s<0$, the path is parabolic; if $E_s=0$, the satellite is in a captive orbit (elliptical, or circular). If $E_s>0$, the path is hyperbolic and the satellite will escape Earth's gravitational pull. The escape velocity is obtained from:

$$(v_{esc}^2 \div 2) - (\mu \div r) = 0; \; v_{esc} = 36{,}700 \text{ ft/sec or about } 12 \text{ km/sec} \quad (4\text{-}4)$$

For launch velocities below v_{esc} , the satellite will either return to Earth (suborbital injection velocities) and follow a ballistic (parabolic trajectory) or orbit in a circular or elliptical orbit with a speed (v) and period (P) determined as below in equation (4-5):

$$P = (2\Pi r) \div v \quad v = \sqrt{\frac{\mu}{r}}$$

Two body gravitational interactions and no energy dissipation are assumed for the present discussion. The effects of solar wind, atmospheric drag and lung-solar perturbation on orbital parameters and decay are discussed in Ch. 7. Figure 4-7 shows the velocity vs. range for a rocket and payload.



(Source: Ref. 2)

FIGURE 4-7. VELOCITY vs. RANGE OF A ROCKET

Since energy is conserved, it is now possible to visualize the exchange between potential and kinetic energy in an elliptical orbit. When the satellite is nearest to the Earth (perigee), the potential energy is least and the kinetic energy is at its peak. Hence the satellite reaches its highest velocity at the perigee and its lowest velocity at the apogee (where the potential energy is highest).

The kinetic energy required for different orbits can be related to a characteristic velocity. The characteristic velocity is also the summation of all the velocity increments attained by propulsion to establish the desired orbit. Table 4-1 (from Ref. 1) describes the characteristic velocities for a number of missions.

TABLE 4-1. MISSION VELOCITY REQUIREMENTS

MISSION	Characteristic Velocity V_{CH} ft/s	Excess Velocity Over Reference Orbit V_{ex}, ft/s
100 nmi Reference Circular Orbit	25,570	0
200 nmi Circular Orbit	25,922	352
500 nmi Circular Orbit	26,900	1,325
1000 nmi Circular Orbit	28,296	2,726
Synchronous Transfer Ellipse	33,652	8,082
Lunar Impact	36,035	10,465
Escape, Parabolic	36,164	10,594
ETR - Synchronous - 28.5° Inclination*	38,490	12,920
WTR - 24 Hour Orbit - "Polar"*	38,503	12,933
Synchronous Equatorial (28.5° Plane Change)	39,791	14,120
Venus Flyby	37,570	12,000
Mars Flyby	37,770	12,200
Mercury (Via Venus Flyby)	38,570	13,000
Jupiter Flyby	36,070	20,500
Saturn (Via Jupiter Flyby)	46,570	21,000
Uranus (Via Jupiter Flyby)	47,270	21,700
Neptune (Via Jupiter Flyby)	50,070	24,500
Pluto (Via Jupiter Flyby)	54,270	28,700
Lunar Landing Return	56,000	N/A

* For an ETR orbit altitude of 19,323 nmi and a WTR orbit altitude of 19,355 nmi.

Angular momentum is also a conserved quantity, so that without an external application of torque for a period of time, a spinning body will neither increase nor decrease its rate of spin. Satellite orbits have an angular momentum, which is about an axis through the center of the Earth. The orbital angular momentum, H, is given by:

$$H = (R)x(v)x(\cos\theta) \qquad (4\text{-}6)$$

where H is the angular momentum, R is the distance from the satellite to the center of the Earth and θ is the angle between the velocity vector and a line in the orbital plane which is perpendicular to the position vector (see Figure 4-8). The product "v x cos θ" can also be referred to as the tangential velocity. H is constant except when the satellite is accelerated or decelerated by thrust or drag.

The equations for conservation of energy and angular momentum are necessary to analyze the dynamics of satellite orbits. The oblateness of the Earth requires some additional terms over those shown in Equation 4-3 for the potential energy expression, to obtain more accuracy in the orbital predictions; and the gravitational fields of the Moon and the Sun, in particular, should also be considered in increasing prediction accuracy.

The plane of the orbit is defined by the longitude of the ascending node and its inclination. These are shown in Figure 4-9. The ascending node is the point where the projection of the satellite path crosses the celestial equator from south to north. The inclination is the angle formed by the plane of the orbit and the equator. It is measured counterclockwise from the eastern portion of the equator to the ascending node. Thus, satellites which orbit west to east (normal or prograde) have an inclination $<90^{o}$; orbits going east to west (retrograde) have an inclination $>90^{o}$. An alternate method sometimes used to designate retrograde inclination is to measure the angle clockwise from the western portion of the equator and state it as an X^{o} retrograde inclination (see Fig. 10-8). A third term often used to describe orbits is the right ascension (ê). This is the arc of the celestial equator measured eastward from the direction of the vernal equinox to the ascending node.

The choice of orbit depends upon the mission of the satellite. ow Earth orbits (LEO) serve a variety of purposes and do not necessarily operate close to the plane of the equator. In fact, orbits with higher inclinations (near polar) provide the satellite the opportunity to cover a larger portion of the Earth's surface(see Figure 4-10).

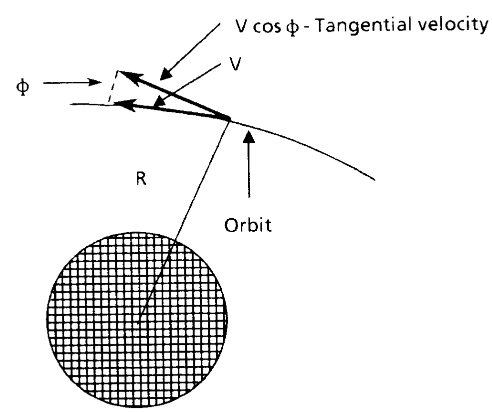


FIGURE 4-8. DEFINITION OF TANGENTIAL VELOCITY

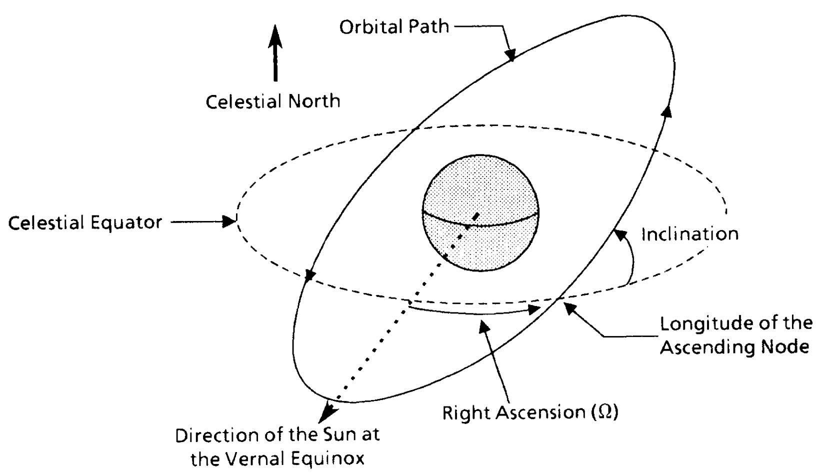
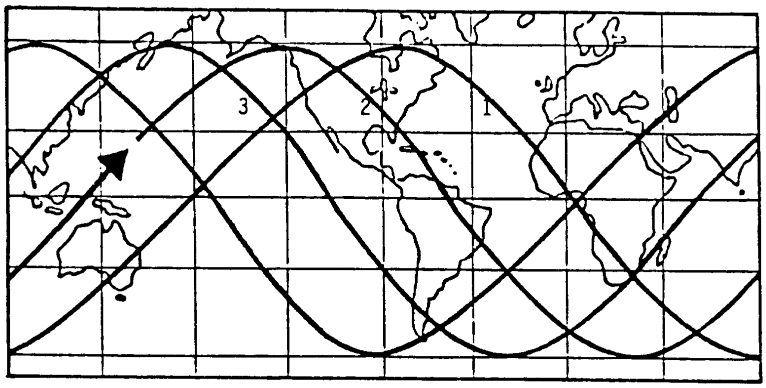


FIGURE 4-9. ORBITAL CHARACTERISTIC DIAGRAM



*Numbers on ground tracks indicate sequence of the orbit, 1 being before 2, etc.

FIGURE 4-10. WESTWARD REGRESSION OF THE ORBITAL GROUND TRACE

Communications satellites are generally placed in geosynchronous Earth orbits (GEO) where they complete one revolution of the Earth in 23 hours, 56 minutes and 4 seconds. A satellite in a geosynchronous orbit on the equatorial plane will appear stationary to observers standing on the equator. In order to have this day-long orbital period,[2] a satellite must be at an altitude of roughly 19,300 nautical miles above the surface of the Earth (5.6 Earth radii). The plane of the orbits of these satellites is either the same as the plane of the equator or at some relatively small inclination angle to the equator. Ideally, equatorial orbits can be achieved directly, with no mid-course

[2] Our "solar day" of 24 hours corresponds to the Earth's apparent spin period but the Earth actually rotates approximately one and 1/365 turns in that time. One rotation of the Earth takes 23 hours, 56 minutes and 4 seconds. Time on a scale based on exactly one rotation of the Earth is referred to as *sidereal time*. One 24-hour day of *sidereal time* is equivalent to 23 hours, 56 minutes and 4 seconds of *solar time*.

corrections, only by launches from the equator. Launches from points north and south of the equator have a minimum inclination which is related to the launch site latitude. Thus, equatorial orbits are normally achieved by maneuvers whereby the satellite is reoriented and a rocket motor is fired perpendicular to the plane of the current orbit to create a new orbit orientation (see Figure 4-11).

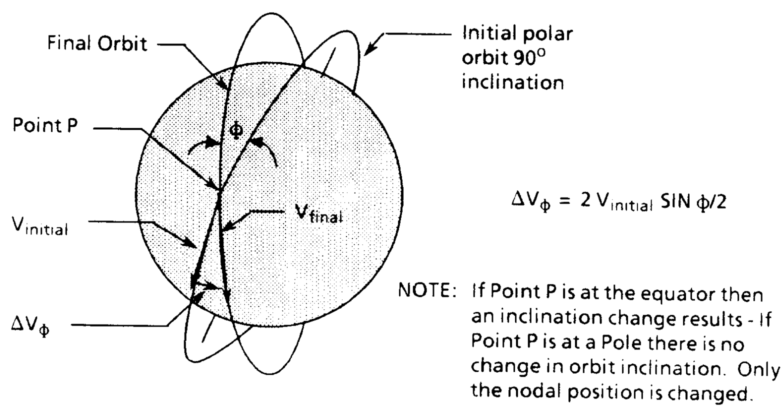


FIGURE 4-11. ORBITAL PLANE CHANGE

Since the orientation of the orbit is relatively motionless in space while the Earth turns inside it, the ground track of the orbit will recess (fall behind). The rate of recession is based on the number of degrees the Earth rotates while the satellite is completing one orbit. The northern-most and southern-most range of the ground track are equal to the inclination of the orbit. A typical ground track is shown in Figure 4-10. The width of the ground track, as seen by the satellite from orbit, is also called a "swath" or "footprint" of the satellite.

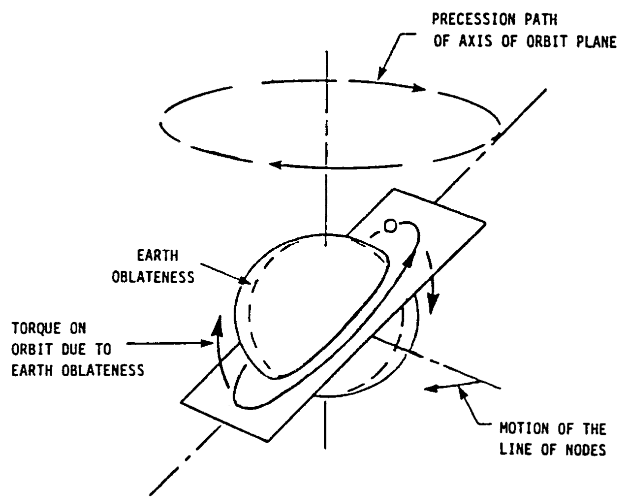
There are external forces which perturb the otherwise stationary orbital plane and cause it to change orientation. The largest effects are caused by the oblateness of the Earth and the gravitational pull of both the Sun and the Moon, called lung-solar perturbations. Their relative importance varies with the altitude of the orbit. The relative effects in terms of acceleration (Earth gravitational units, or g's) for a satellite 200 n mi. above the Earth are shown in Table 4-2. As the altitude of the orbit increases, the relative effect of the Earth's oblateness decreases and the Sun and the Moon's influence increases.

TABLE 4-2. COMPARISON OF RELATIVE ACCELERATION (IN G's) FOR AN EARTH SATELLITE AT 200 NM ALTITUDE (2)

Source of Perturbation	Equivalent Acceleration (in g's) on 200 n mi. Earth satellite
Earth's attraction	0.89
Earth's oblateness	0.001 (approx.)
Sun's attraction	0.0006
Moon's attraction	0.0000033

While the attraction of bodies other than the Earth can distort the orbit, the oblateness of the Earth will cause the plane of the orbit to precess around an axis through the pole of the Earth. The additional girth of the Earth around the equator (oblateness) produces a torque on the orbit and the result is a precessional motion not unlike that of a gyro or top. The precession rate can be defined as the number of degrees the line of nodes moves in one solar day. The nodal precession rate for circular orbits is shown in Figure 4-12. Note that the effect of the Earth's oblateness lessens with the altitude of the orbit and also with the inclination of the orbit. A polar orbit will not precess.

The rotation of the Earth has an influence on the ability to launch satellites into desired final orbits. Looking down upon the North Pole, the Earth rotates counterclockwise. At the time of launch, the rocket already has a horizontal component of velocity which equals in magnitude the product of the Earth's rate of rotation and the distance to the axis through the poles of the Earth. If the ELV is launched in the direction of this velocity vector (eastward), it reaches orbital velocity easier than if it is launched in a westerly direction, in which case this surface velocity must be first overcome. (This effect varies with the latitude of the launch point. It is greatest at the equator and absent at the North or South Poles.) This factor is one influence on selection of a site for conducting launches. Therefore, in the United States, eastward launches of satellites into equatorial orbits from ETR, Florida augment the ELV thrust. More payload can be placed into orbit than from an identical launch made from, for example, Maine. The satellite launches from the West coast are almost always to the south to achieve polar (high inclination angle) orbits. Polar orbits are perpendicular to the velocity provided by the Earth's rotation, thus the rotation neither helps nor hinders the polar launch. However, the launch corridors used at both ETR and the West coast are chosen primarily for safety considerations. Launches eastward from ETR and southward from the West coast fly over water rather than inhabited territory and do not pose hazards to populated areas due to jettisoned stages or other debris.



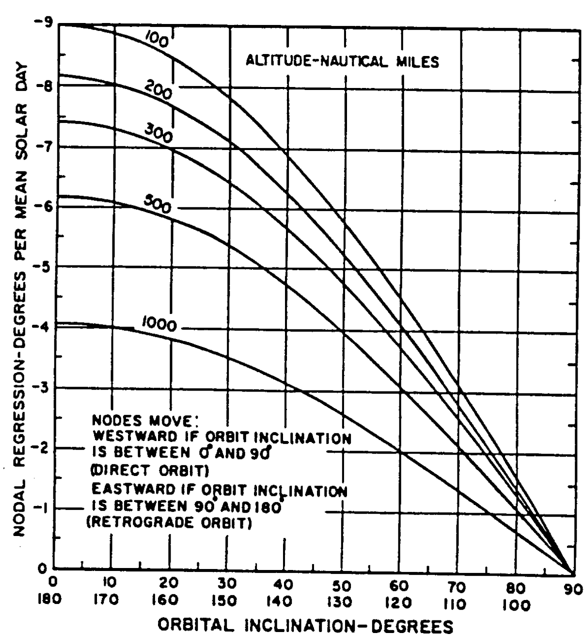


FIGURE 4-12. ORBITAL PLANE PRECESSION DUE TO EARTH'S OBLATENESS (Ref. 2)

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